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WIND TUNNEL DYNAMIC FLYING STUDY OF THE PITCHING MOMENT DERIVATIVES OF THE STANDARD DYNAMICS MODEL IN ACTIVE CONTROL

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Abstract

This paper presents the results of a dynamic wind tunnel flying study of the pitching moment coefficients of the Standard Dynamics Model as a function of angle of attack ranging from about -10 deg to +30 degrees. The fully instrumented model with its servoed elevator has been flown in a single pitch degree of freedom at low speeds under moment trim conditions covering both the stable CG locations of 19.2% ,25% and the unstable CG location of 35% of the mean aerodynamic chord. In the unstable case an active control law has been invoked to fly the model in an angle of attack control mode . The freely flying model has been disturbed with flight test like inputs and the responses have been analysed to generate the aerodynamic pitching moment coefficents using maximum likelihood estimation procedure. The analysis includes the closed loop active controlled system parameter estimation and the results are compared with the available SDM pitching moment data.

Nomenclature

b_x	Bias errors
$\bar{c}$	Mean Aerodynamic Chord (MAC), m
$C_{m\alpha}$	$\partial C_m/\partial\alpha$ or $M_\alpha I_y / \bar{q} S \bar{c}$,rad^{-1}
$C_{mq} + C_{m\dot{\alpha}}$	$\partial C_m/\partial(q \bar{c}/2U) + \partial C_m/\partial(\dot{\alpha}\bar{c}/2U)$, rad^{-1} or $2 U I_y (M_q + M_{\dot{\alpha}})/ \bar{q} S \bar{c}^2$
$C_{m\delta}$	$\partial C_m/\partial\delta$, $M_\delta I_y / \bar{q} S \bar{c}$, rad^{-1}
q	Pitch rate, rad/sec
M_x	Dimensional pitching moment coefficients
α	Angle of Attack, rad
δ	Elevator angle. r
δ_c	Elevator Command,rad
N	Number of samples used for analysis
$\bar{q}$	Dynamic pressure. Pa
X	Math Model state vector (q and $\dot{\alpha}$)
Y	Computed Observation Vector (Variables with m subscript)
U	Tunnel Flow Velocity, m/sec
Z	Measurement vector (Z = Y + n) n, measurement noise vector
R	Noise Covariance Matrix
I_y	Mass moment of Inertia of the model about the y axis through CG, Kg m^2
K_1	Angle of Attack feedback factor for active control, rad-elevator/ rad-alpha
K_2	Pitch rate feedback factor for active control, rad-elevator / rad/sec-pitch rate
t	Time. sec
($\dot{}$)	First Derivative with respect to time
($\tilde{}$)	Small perturbation around a mean

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Introduction

The stability derivatives of scaled aerodynamic models are usually extracted from wind tunnel experiments by analysis of the force data either from forced oscillation rigs or from articulated motions of the model in wind tunnels [1] . For aircraft in flight, stability derivatives are usually extracted from the analysis of flight test responses of the vehicle to specific control surface inputs, using Parameter Estimation techniques [2]. Dynamic wind tunnel flying experiments combine features of both these techniques ,namely creating the characteristic free motion responses of the model to test inputs under trimmed conditions in the wind tunnel environment. No force measurements are made. The motion responses are analysed using Parameter Estimation techniques.Therefore dynamic wind tunnel techniques allow flight test like investigations to be conducted in wind tunnels.

Semi-free dynamic wind tunnel flying of models have been reported in the literature ,and the method consists of supporting the model in a wind tunnel with some rotary and translational freedom [1,2]. Typically in such studies, the model control surfaces are servoed for controlling the model attitude and are instrumented for measuring the model motion responses.

A literature survey reveals that in respect of a single degree of freedom flying a basic finner model has been flown with pitch freedom to evaluate the pitching moment characteristics of the model at supersonic mach numbers using log decrement analysis [3]. A Do-28-INI scaled model has been flown with three degrees of freedom to evaluate the control characteristics in a control mode, and the responses have been analysed to generate the stability derivatives of the model using parameter estimation methods based on maximum likelihood estimation (MLE) procedure [4]. With fully servoed control surfaces, active control of statically unstable models in semi free flight is feasible, but control of unstable models in dynamic wind tunnel flying for purposes of control law evaluation have not been reported in the literature. The dynamic wind tunnel flying methods hold good potential for stability derivative estimation, modeling as well as for control evaluation and need to be explored. This paper is one such effort.

The paper presents results of a simple experimental study for dynamically flying aerodynamic models in a single degree of pitch freedom in active control for evaluating the pitching moment behavior of the model over realtively large range of trimmed angles of attack. The study is around the Standard Dynamics Model (SDM) so as to validate the experimental and analysis technique. The experimental study has been made for both statically stable and unstable Center of Gravity (CG) locations. In the case of the unstable CG location, the model motion in the tunnel flow has been stabilised by invoking an active control law. The study thus aims to demonstrate the wind tunnel dynamic flying to be potentially viable method for routinely generating moment stability derivatives of the aerodynamic model and for control law evaluation.

Pitching Moment Math Model and Active Control

Consider an aerodynamic model with a free pitch hinge at its CG mounted in the test section of a wind tunnel. The three forces and the roll/yaw moments acting at its Center of Gravity under the flow conditions are reacted by the rigid pitch hinge and the pitching moment induces free pitching motion of the model which is representative of its moment characteristics. This pitch motion can be expresssed as [5]

$$\dot{q} = M_{\alpha}\tilde{\alpha} + (M_q + M_{\dot{\alpha}})\, q + M_{\delta}\tilde{\delta} \qquad (1)$$

where, '$\dot{q}$' is the model pitch accelaration, 'q' is the pitch rate of the model. α is the angle of attack, $\tilde{\alpha}$ is the small perturbation of the angle of attack around a trimmed equilibrium point, δ is the elevator angle and $\tilde{\delta}$ is the small perturbation to the elevator around a trimmed equilibrium angle. In this representation pitch angle is taken to be the angle of attack. The M_x coefficients correspond to the dimensional pitching moment coefficients of the model for the given angle of attack. Generally the aerodynamic moment coefficients of most models can be expected to be nonlinear with angle of attack, and hence equation 1 can be considered valid only for small perturbation around a trim point corresponding to condition C_{mcg} =o. The damping terms of the equation 1 correspond to pitching motion of the aerodynamic model around a fixed axis. It is the sum of the rotary and the translatory componencts which physically occur at the same time even though the two variables $\dot{\alpha}$ = q are equal, their damping effects are different. Trim can exist for many combinations of angle of attack and elevator position , the locus of which corresponds to pitch trim curves.

For the statically stable CG locations, M_{α}<0, and for positive damping, the model pitching motion in free flow is convergent and inherently stable. On the other hand, for CG locations which makes the model statically unstable, M_{α}>0, the pitching motion is divergent and it is necessary to stabilise pitching motion under flow conditions by feedback. to obtain angle of attack control. This is precisely the problem encountered in the design of active control for modern unstable aircraft. The angle of attack loop stabilisation can be realised by feedback of the model states to a high response elevator servo drive.

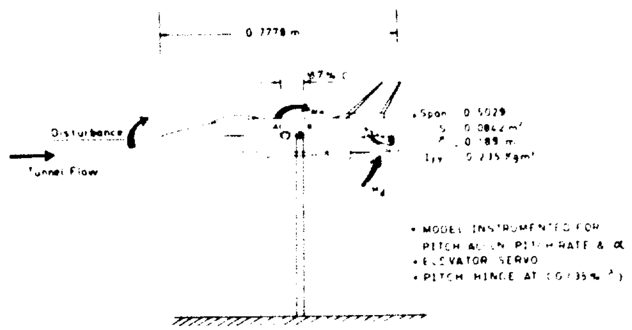


Fig. 1 Unstable Model Dynamic Wind Tunnel Flying Scheme

Figure 1 shows a schematic block diagram for dynamically flying statically unstable model in pitch freedom using an active feedback control law based on angle of attack (since the instability is an angle of attack based one) and the pitch rate. The angle of attack feedback serves to make the effective M_{α}<0 for the loop. The figure 2 shows active control schematic block diagram dynamic for a statically unstable CG location.

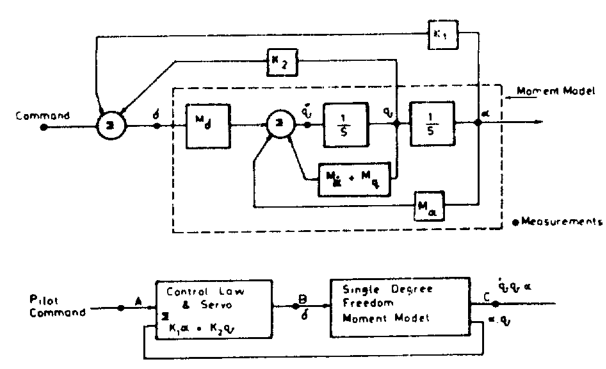


Fig.2 Active Control Schematic Diagram

From the control schenic it can be noted that the angle of attack feedback factor K_1 needs to be higher than ratio M_{α}/M_{δ} for the loop to be stable and the K_2 feedback improves the pitch damping. This pitching moment math model assumes ideal dynamics for the control surface drive and for the pitch rate / angle of attack sensors.

Dynamic Wind tunnel Experiments

The Standard Dynamics Model was fabric-red using machined laminates for the wings,the horizonta, tail ,the vertical tail and the vertical fins. The aluminium uselage was made hollow by using shcetmetal constructio: The actuator with the low backlash gearing system,the sensors, and the pitch hinge mechanism were housed in the hollow fuselage. The pitch hinge articulation mechanism consisted of a pair of ball bearings rigidly fixed to the body such that the center of rotation was precisely set to the *CG* of the model. Three locations corresponding to 19.2% ,25% and 35% of the mean aerodynamic chord were chosen for the *CG* locations for different sets of experiments. In each case lead weights were redistributed to adjust the three dimensional *CG* precisely on the pitch hinge axis.

The horizontal tail consisted of the pair of tail elements set at a dichedral angle of -10 deg and was driven by a worm wheel/worm gear box coupled to a permanent magnet DC motor. This drive was in a position control mode through an elevator position sensor and had a capability of 8Hz flat rcpsonse for small perturbation. The slew rate capability was better than 300 deg/sec. The control repeatability was better than 0.1 deg and the elevator had a range of movement of +/_ 25 deg.

The model was instrumented with a +/_2g strain gauge accclarometer having a flat response upto 50Hz and was located at a typical distance of about 300 mm from the CG to serve as a pitch acceleration sensor. A 400 Hz supply driven rate gyro of 60 deg/sec range was used to sense the pitch rate, and the output was demodulated by a phase sensitive detector. The position potentiometer on the elevator and a wind vane type potentiometric angle of attack sensor were also mounted within the fuselage. The various electical devices were cabled out as a flexible catenary to a work station outside of the tunnel. This station housed the video monitor showing the model, the angle of attack display and the stick/trim control with doblet generator for elevator. An analog computer was used to generate the real time control law for the statically unstable model dynamic flying. The unfiltered model motion data was taken to a PDP-11 computer Tor data acquisistion purposes and were sampled at a rate of 50 samples/sec. Both the openloop and active control experirnents were conducted from this work station. The sensors were presicely calibrated and the moment of Inertia I_{yy} was determined on an inertia measuring rig with the model CG properly located.

The **SDM** was mounted in a **1.2m X 1.2** m low speed tunnel run at a fixed wind velocity of 32 m/sec and was dynamically flown in the pitch degree of freedom. The test Reynolds Nurnber was about 0.5 million/chord. With the wind tunnel operating, for the stable CG locations of **19.2%** and **25%** of MAC, the model sought a pitch trim ranging from -10 deg to +30 deg angle of attack depending upon the elevator setting. At each trimmed angle of attack, the model was disturbed by a small amplitude zero mean doublet of about 600 millisecond duration to the elevator to realise an angle of attack disturbance of no more than +/_2 deg . A set of runs were conducted in open loop covering angle of attack range of -10 deg to about + 30 deg in about **IS** steps.

For the unstable CG location of **35% MAC**, the **SDM** pitching motion was inherently divergent and could be stabilised by use of an active control feedback discuss~~ed~~ in the previous section and shown schematically in figure-2. Initially, the control law was tried with a finite K_1 and K_2 **set at** zero. The model went into oscillations and could be stabilised by use of K_2 feedback.In a second experiment. K_1 was gradually reduced and the model diverged at a finite positive value of K_1. The control law used was,

$$\delta = \delta_c + 0.33\,Q + 0.045\,q + \text{bias} \qquad (2)$$

The above control law was chosen so as to have a loose control on the angle of attack so that while estimating the aerodynamic coefficients from the small perturbation response,the loop closure was not vsy strong. Again a set of runs covering -10 deg to about +30 deg were conducted and the doublet response results were recorded.

Method of analysis

Identification and Parameter Estimation methodology is now a mature field of analysis with many validated codes which can handle both linear and nonlinear models and observation noise [1,2]. These methods use the input-output reponses of physical systems to known inputs to estimate the parameters of the math model representing the physical system. In the present study, a maximum liklihood estimation procedure has been used to evaluate the aerodynamic pitching moment coefficients from the input output reponses of the dynamically flown model. The mathematical model used to represent the motion of the aerodynamic model for small perturbation around the trim point has been discussed in equation I. This math model is expressed as state and observation equations and includes control term ,measurement vector , and noise terms for puposes of parameter estiamtion as follows,

State equations of pitch motion

$$\begin{aligned} \dot{q} &= M_\alpha \tilde{\alpha} + (M_q + M_{\dot{\alpha}})\, q + M_\delta \tilde{\delta} + b_1 \\ \dot{\alpha} &= q + b_2 \end{aligned} \qquad (3)$$

Observation equations of pitch motion

$$\begin{aligned} \dot{q}_m &= M_\alpha \tilde{\alpha} + (M_q + M_{\dot{\alpha}})\, q + M_\delta \tilde{\delta} + b_3 \\ q_m &= q + b_4 \\ \alpha_m &= \alpha + b_6 \end{aligned} \qquad (4)$$

In the above model. the state and observation equations have been written with bias parameters b, to b_6. The control input in this set of equations is δ ,The model as in equations 3 and 4 above have eight unknown parameters. The three pitching moment coefficients include thr single damping term accounting for both the pitch rate rotary term and the translatory $\dot{\alpha}$ term. The measured variables were converted to SI units prior to estimation and the three pitching moment coefficients correspond to the dimensional pitching moment stability and control derivatives.

The concept behind maximum liklihood estimation is that for each possible estimate of the unknown model parameter,the probability that the model motion response history attains trajectory very near the observed trajectory can maximised. Mathematically,the maximum liklihood estimation procedure aims at minimising the negetive of log liklihood function,

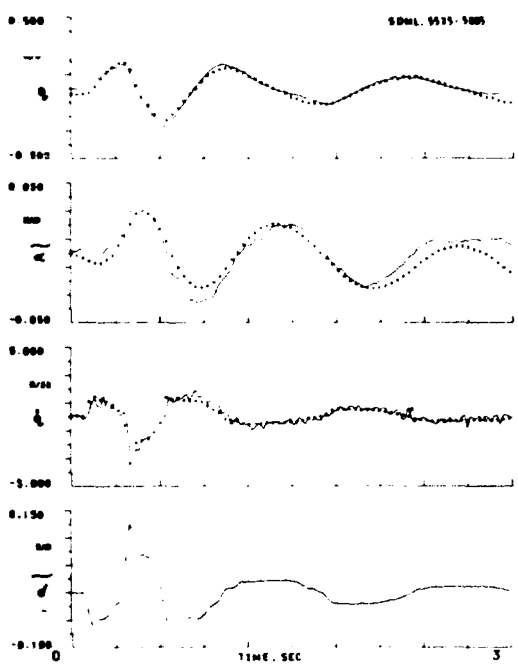
$$L = 1/2 \sum_{I}^{N} [Z(t) - Y(t)]^T R^{-1} [Z(t) - Y(t)] + N/2\, ln\, |R| + \text{Const} \qquad (5)$$

where **Z** is the measurement vector, **Y** is the computed observation vector, N is the number of sampled sets of input-output time histories. The analysis assumes that the measurements are corrupted by a zero mean gaussian noise n and R is the measurement noise covariance matrix. Typically a record of length of about 3 to 6 seconds corresponding to an N ranging from 200 to 400 was chosen for analysis The software is based on a quasilienarisation method,equivalent to modified Newton-Raphson method and uses an approach which neglects the computation of the second gradient of error [6]. It evaluates the parameters of equations 3 and 4 and prints out the goodness of fit in terms of the standard deviation of the parameters. An overlay of the actual time response on the estimated response based on the parametr estimation is also generated.

The model responses at the various trimmed angles of attack were analysed using MLE software. In the case of the openloop runs. recorded input-output pairs were used with the doublet at elevator representing the input. In the case of the active control experiments. the chosen input-output were from points B snd C of the figure-2, with input corresponding to the actual elevator position. The results of the analysis are discussed in the following section

Result; and Discussions

The results of the MLE analysis indicated that the chosen moment model for describing the pitching motion was very good for the statically stable CG location experiments.This was based on the typical standard deviation of the model parameters and these were better than 1 to 2 %. The response matching was also very good. In the case of the statically unstable CG location experiments, the standard deviation of the moment coefficients was in the range of 5 to 7% The estimated and the actual time responses are compared, for the 35% CG location. in figure-3. The MLE analysis runs for the unstable CG case with active control did pose some problems of convergence and needed proper initial conditions to converge,unlike the stable runs.



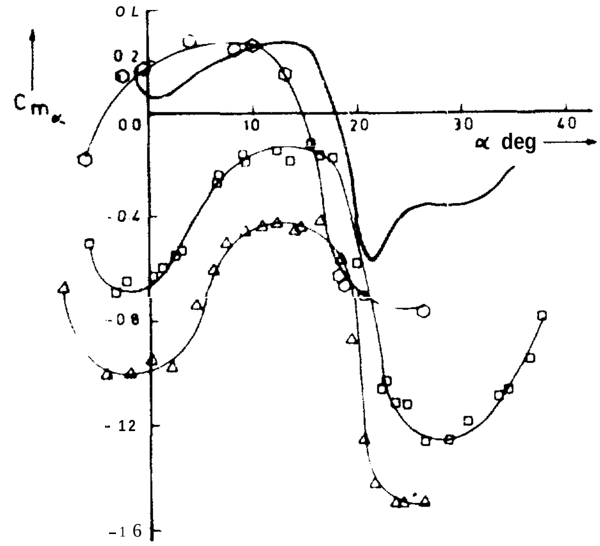
———— Measured Response
++++++++ Estimated Response

Fig. 3 Curve Fits from Parameter Estimation

The non-dimensional body axis pitching moment coefficients $C_{m\alpha}$, $(C_{mq}+C_{m\dot{\alpha}})$, and $C_{m\delta}$ and the trim curves are shown in figures 4 to 8, fur the three CG locatinns

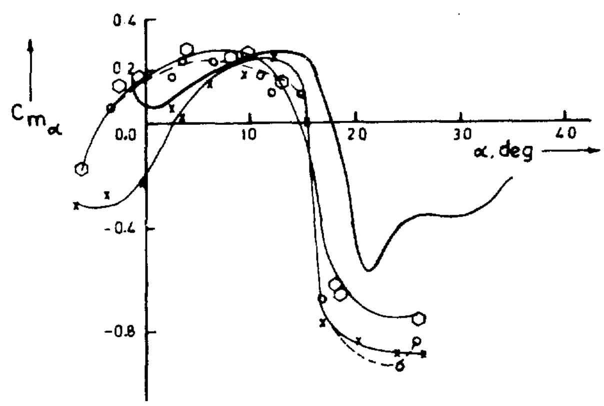
Static Stability Vs Trimmed angle of attack

Figure 4 shows a plot of static stability as a function of trimmed angle of attack fur 19.2%, 25% and 35% locations of CG It can be noted that the plots provide Static Margin (SM) variation as a function of trimmed angle of attack The plots suggest that the SDM model has most forward neutral point in the angle of attack range of about 5 to 15 deg angle of attack



SYMBOL	CG LOCATION	TEST	ANALYS'S (FIG-2)
△	19.2 % $\bar{c}$	Open loop	BC
□	25 % $\bar{c}$	Open loop	BC
○	35 % $\bar{c}$	Active Control	BC
—	35 % $\bar{c}$	REF- 7	

Fig. 4 Static Stability Vs Trimmed Angle of Attack CG at 19.2%, 25%,& 35% M A C Location



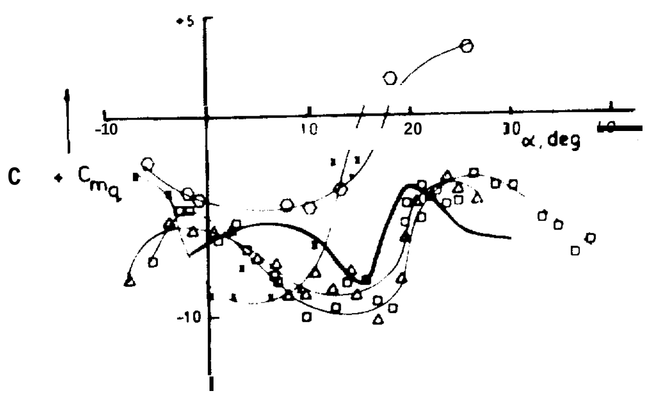
SYMBOL	CG LOCATION	TEST	ANALYSIS (FIG 2)
○	35 % $\bar{c}$	Active Control	BC
o	35 % $\bar{c}$	Active Control	BC
x	35 % $\bar{c}$	Active Control	AC / Corrected for BC
—	35 % $\bar{c}$	REF- 7	—

Fig. 5 Static Stability Vs Trimmed Angle of Attack CG at 35% MAC Location

The three plots of $C_{m\alpha}$ for tlic various CG locations run roughly parallel to each other. It canbe noted that the shift in $C_{m\alpha}$ caused by shift in CG location .in the range of angle of attack between 5 to 15 deg, suggests on the average a $C_{L\alpha}$ of approximately about 4, based on ttic identity $C_{m\alpha} = C_{L\alpha} \times SM$, where SM is the Static Margin. The figure also shows plot of $C_{m\alpha}$ as a function of angle of attack from AEDC [7]. This result is for a **SUM** like model and is from a forced oscillation test at mach number **of** 0.2 and is for a fixed elevator setting of **-5** deg. The result corresponds to 35% CG location. The results from tlic present study in respect of 35% CG location generally matches results from AEDC for a similar configuration [7]. The study duplicates the sharp pitch down **in** the neighborhood of **IS** to 18 deg angle of attack lor all CG locations. However,at the CG location of 35% **MAC.** the results from the present study indicates the pitch down to occur a few degrees in angle of attack earlier and there are differences at small angles as well as beyond 20 deg angle **of** attack. When ttic results from present **study** are compared with SDM data from FFA [8](data being for Mach number of 0.6), the general pattern of instability from about 2 deg to 15 deg is repeated in both. Small differences between ref [7,8] and present study could be due to fact that the dynamic flying experiment was conducted under pitch trim condition5 with elevator position varying as a function of steady state trimmed angle of attack .

Figure 5 shows results for two sets of repeat runs and tuo different kinds **of** analyses. The first used signals between B and C of figure-2 as recorded ignoring loop closure and possible correlations . Since the control was in closed loop, tlic signals at **A** anti C could be correlated giving rise to errors in estimation.In the second type of analysis, the signals pickedoff at points A **and** C and,were analysed using the same MLE procedure to generate tlic closed loop moment coefficients. Since **the** control law was precisely known, an evaluation **of** $C_{m\alpha}$ **of** the basic SDM was made. Ttic figure 5 indicates general agreement amongst the three results for various CG locations with some differences at low angle of attack. The results indicate that the SDM is about 7% unstable between **5** and 15 deg angle of attack.

Pitch damping Vs Trimmed angle of attack



SYMBOL	CG LOCATION	TEST	ANALYSIS (FIG-2)
Δ	19.2 % $\bar{c}$	Open loop	BC
□	25 % $\bar{c}$	Open loop	BC
○	35 % $\bar{c}$	Active Control	BC
x	35 % $\bar{c}$	Active Control	AC / Corrected for BC
—	35 % $\bar{c}$	REF - 7	—

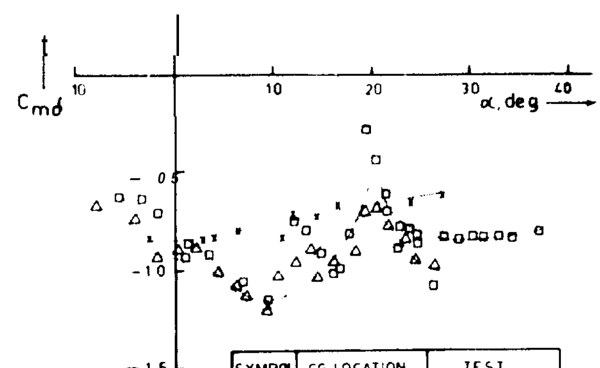
Fig. 6 Pitch Damping Vs Trimmed Angle of **Attack**

In figure 6, a plot of the pitch damping for various CG locations as estimated from tlic MLE analysis is shown. In **respect of** the openloop experiments for the stable CG locations **of** 19.2% and 25% of MAC, tlic damping pattern is very similer to the results reported from AEDC [7]. However, for the active control runs corresponding ot the CG location of 35% MAC, the pattern of damping is different suggesting areas of negative damping beyond 15 deg trimmed angle of **attack.** The results from FFA [8] for SDM model at mach number of 0.6 docs show zones of very low to negetive pitch damping.

The usual definition of pitch damping $C_{m_q} + C_{m_{\dot\alpha}}$ is based on **a** fixed wing -tail geometry **accounting** for both the translatory and the rotary motions **of** the model in the fluid flow. In the case of active controlled experiments, it may be noted that the elevator position is continuously varying relative to the uing over the period of control , where as in the stable CG case the elevator position is fixed,at trimmed value after the input doublet ,during the model response perrod. This geometrical difference could account for the different damping pattern between the stable and unstable flyingexperiments. . On the whole , the damping results from the present dynamic flying study generally matches the damping results from other sources also.

Control effectiveness Vs Trimmed angle of attack

In figure-7 , the control effectiveness is shown as a function of trimmed angle of attack for various CG locations.



SYMBOL	CG LOCATION	TEST
Δ	19.2 % $\bar{c}$	Open loop
□	25 % $\bar{c}$	Open loop
x	35 % $\bar{c}$	Active Control

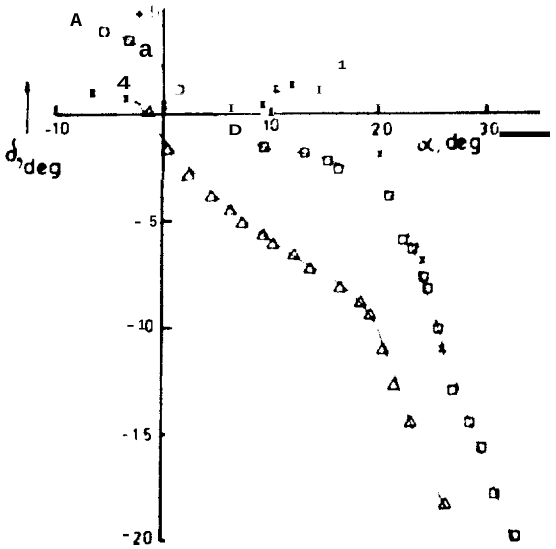
Fig. 7 Control Derivative Vs Trimmed Angle of Attack

For the two statically stable CG experiniciits conducted under openloop conditions. tlic control effectiveness match. Under closed loop active control, the control effectiveness has **a** different character. The plots clearly **show a** singularity at the in tlic angle of attack range 15 to 18 deg at which Sharp pitch down in $C_{m\alpha}$ occurs. This data is useful in active control system design and in handling quality studies.

Elevator Vs Trimmed angle **of** attack

Figure-8 **shows** the trim curves **for** the tlirce CG locations. The plots indicate tlirce zones **of** different gradients suggesting nonlinear $C_{m\alpha}$ as a function of angle *of* attack. The slope of the trim curves correlate *to* the static stability $C_{m\alpha}$ showing lowest gradients in the **5** to 15 deg angle of attack range. In the case CG location of

35% MAC, the slope of the trim curve is reversed from about nearly zero to 17 deg angle of attack. The slight scatter in the trim curve in respect of 35% MAC CG location in the unstable zone is due to continuous activity of the tail in active control and hence the uncertainty in the mean value. The trim curves are consistent with the typical aircraft charteristics.



SYMBOL	CG LOCATION	TEST
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Fig 8 Trim curves for CGs at 19.2%,25% & 35% MAC

Conclusions

This study has demonstrated the potential of wind tunnel dynamic flying techniques in stability and control evaluation of scaled aerodynamic models. The limited degree of freedom dynamic flying technique is based on generating, analysing and controlling free model motion with a limited degree of freedom in a wind tunnel. Specifically, the paper has presented results of a single pitch degree of freedom study to generate pitching moment derivatives of the Standard Dynamic Model by a flight test like experiment within a wind tunnel. Secondly, the study has detailed an active control experiment in which the nearly 7% statically unstable model is flown in an angle of attack mode invoking an active control law. The pitching moment derivatives of the model have been evaluted both under open loop and closed loop conditions using proven maximum likelihood parameter estimation method. The result and the technique has been validated by reasonable comparisons with derivative data from other sources.

This study suggests that the dynamic wind tunnel flying experiments route is a simple and viable technique for routine generation of aerodynamic stability data in wind tunnels and is a tool for studying active control laws. Though the present study was limited to a single degree of rotational freedom, the technique is directly applicable to the three rotational degrees of freedom with trimming features in yaw and possibilities in generating coupling derivatives.

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$C_{mq} + C_{m\dot{\alpha}}$	$\partial C_m/\partial(q\bar{c}/2U) + \partial C_m/\partial(\dot{\alpha}\bar{c}/2U)$, rad^{-1} or $2 U I_y (M_q + M_{\dot{\alpha}}) / \bar{q} S \bar{c}^2$
$C_{m\delta}$	$\partial C_m/\partial\delta$, $M_\delta I_y / \bar{q} S \bar{c}$, rad^{-1}
q	Pitch rate, rad/sec
M_x	Dimensional pitching moment coefficients
α	Angle of Attack, rad
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Introduction

The stability derivatives of scaled aerodynamic models are usually extracted from wind tunnel experiments by analysis of the force data either from forced oscillation rigs or from articulated motions of the model in wind tunnels [1] . For aircraft in flight, stability derivatives are usually extracted from the analysis of flight test responses of the vehicle to specific control surface inputs, using Parameter Estimation techniques [2]. Dynamic wind tunnel flying experiments combine features of both these techniques ,namely creating the characteristic free motion responses of the model to test inputs under trimmed conditions in the wind tunnel environment. No force measurements are made. The motion responses are analysed using Parameter Estimation techniques.Therefore dynamic wind tunnel techniques allow flight test like investigations to be conducted in wind tunnels.

Semi-free dynamic wind tunnel flying of models have been reported in the literature ,and the method consists of supporting the model in a wind tunnel with some rotary and translational freedom [1,2]. Typically in such studies, the model control surfaces are servoed for controlling the model attitude and are instrumented for measuring the model motion responses.

A literature survey reveals that in respect of a single degree of freedom flying a basic finner model has been flown with pitch freedom to evaluate the pitching moment